CRYOGENIC AND LOX BASED PROPULSION SYSTEMS FOR ROBOTIC PLANETARY MISSIONS

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ABSTRACT

Robotic planetary missions use almost exclusively storable propellants. However, it is clear that the use LOX / LH2 and LOX / HC combinations will offer a tremendous payload gain for most robotic missions.
The perceived complexity of cryogenic propulsion, the need of cryogenic propellants active refrigeration to eliminate boil-off losses and the high acceleration imparted to spacecraft structure by high thrust cryogenic engines have prevented the use of cryogenic propulsion for robotic missions.
Elaborated thermal insulation techniques and the recent progresses in active refrigeration help to consider the use of cryogenic propulsion for interplanetary missions.
The performance increase enable to consider new missions like Mars sample return without in orbit rendezvous or heavy rovers on Mars surface.

1. INTRODUCTION

Robotic planetary missions are - up to now - using almost exclusively storable propellants.
Despite their higher specific impulse, the LOX based combinations are not considered owing to the propellant loss by cryogens evaporation.
However, it is clear that the use LOX / LH2 and - to a lesser extent – of LOX / HC combinations will offer a tremendous payload gain for most robotic missions.
Three points have prevented the use of cryogenic propulsion for robotic missions:

- The perceived complexity of cryogenic propulsion.
- The high thrust of existing main engines, leading to excessive stresses on spacecraft appendages.
- The need of active refrigeration to eliminate cryogenic propellants boil-off losses.

However, in the recent years, cryogenic refrigerators made impressive progresses in reliability and efficiency. Combined with an efficient superinsulation, this enables to store cryogens in space for indefinite periods, provided electrical power is available.
In addition, a new concept "Low Cost Cryogenic Propulsion" (LCCP) enables to drastically simplify the propulsion stage (tanks self pressurisation, low pressure engine, elimination of chill down sequence, multiple restarts) and to reduce acceleration to the level provided by ABM, i. e. around 0.1 m/s2 . This level is compatible with deployed solar panels and antennas.
It is thus perfectly possible to plan heavier automatic missions with cryogenic propellants.

2. LOW COST CRYOGENIC PROPULSION (LCCP)

2.1. Concept description

A new concept "Low Cost Cryogenic Propulsion" (LCCP) enables to drastically simplify the propulsion stage (tanks self pressurisation, low pressure engine, elimination of chill down sequence, multiple restarts). The thrust level (500 N to 2000 N) is sufficient to provide an acceleration of 10-2 m/s^2 (identical to the acceleration provided by an ABM (Apogee Boost Motor) to a geostationary satellite). This acceleration is sufficient for most planetary manoeuvres (e. g. planetary capture). The schematic diagram of LCCP is shown on figure 1. Propellant tanks operating in microgravity, attitude control concept and thruster deserve more detailed description.

2.2. Propellant tanks

The tanks are insulated by hybrid insulation: foam for ground operation and superinsulation for space operation. Associated to low thermal conductivity mechanical interfaces (CFRP struts); this enables to reduce the heat flux received by the tank below 150 W for Earth Orbit applications. This

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is the case when LCCP is used only for the first step of the interplanetary mission, injection into interplanetary orbit.

The heat input (150 W) is less than the average power required to maintain the tank pressure versus increased ullage volume by propellants evaporation. In other words, the low thermal flux is an intrinsic protection against pressure surges. For missions involving long storage duration in space, it is advisable to reduce further the heat flux (one order of magnitude) in order to keep the active refrigeration mass to a reasonable level.

Propellant acquisition is performed by a simplified PMD with a buffer volume, combined with local cooling (e. g. by propellant evaporation through a Joule Thomson plug or by the cold head of a refrigerator) in order to insure that liquid is always present at the tank outlet. Thermal gradient inside tank is controlled by propellant circulation, for example through a spray tube.

Tank pressurisation:

Tank pressurisation is due in part to the heat input. Propellant evaporation inside a heat accumulator provides the supplementary gas flow. By this way no helium pressurisation tank is required.

Due to the thermodynamic properties of hydrogen, it is more effective to retain a low pressure on hydrogen side (approx. 0.2 MPa). So a small electric pump is used to raise the pressure to 0.75 MPa to feed the main engine.

2.3. Attitude control

During propulsive phases, the main thrusters insure pitch and yaw.

The figure 1 depicts a solution with four fixed thrusters, using thrust modulation for attitude control. A single thruster mounted on gimbals could be also used.

The roll control is insured by auxiliary thrusters (GOX GH2 on the figure). Heated GH2 could be also used (simpler solution but requiring a bigger H2 heat accumulator).

During non-propulsive phases, the RCS is controlling the three axes in a coarse mode to maintain solar panel pointing toward the sun and to cancel the effect of perturbations (gravity gradient, drag, solar radiation pressure).

2.4. Main engine

The main engine has a thrust comprised between 500 N and 2000 N.
The 500 N engines has been studied in more details.
It offers the following characteristics:
Regeneratively cooled chamber, radiatively cooled nozzle extension.

Chamber length	= 140 mm
Throat diameter	= 24 mm
Nozzle area ratio	= 200
Nozzle exit diameter	= 340 mm
Overall length	= 640 mm
Mixture ratio	= 5 to 6
Specific impulse	= 440 s (for a theoretical value of 482 s)

The injection is performed with gaseous hydrogen and LOX for optimum stability.

The main engine can be fired several hours and sustain more than 40 starts / stop cycles.

The flowrate is sufficiently small to enable electrical pump feeding or pressurisation by propellant evaporation.

The 2000 N engine being larger, the specific impulse may be slightly higher. This engine needs a two axes steering mechanism to insure attitude control.

3. CRYOGENIC PROPELLANTS ACTIVE REFRIGERATION

In the recent years, cryogenic refrigerators made impressive progresses in reliability and efficiency. Off the shelf space qualified cryorefrigerators are now available in Europe for the 90 K range (LOX).
The elaborated tanks thermal insulation (foam and MLI superinsulation on walls, low thermal conductivity thermal struts, tank in solar panel shadow, V grooves) enable to reduce the in-orbit thermal flux to 50 W on the LOX tank and 15 W on the LH2 tank (5 tons propellant). The tanks radiative insulation is shown on figure 3. A moderate solar panel power (1500 W) is sufficient at 1 AU to provide the cryorefrigerator supplementary power.
Such an LH2 propulsion module will enable to put more than 7 tons on Mars orbit, thus allowing for example to perform a Mars sample return mission with direct return to Earth without Mars orbit rendezvous.

3.1. LH2 / active refrigeration example

- Heat losses to LH2 tank (propellant mass = 800 kg) are reduced to 10 W in space by superinsulation (foam insulation during atmospheric phase), and CFRP truss holding frame.
- The use of LOX tank as an intermediate 90 K shield will enable to further reduce heat losses.
- With convenient shielding, the LOX tank can be almost controlled by passive thermal control in interplanetary orbit (figure 2 and 3).
- The corresponding electrical power will be less than 1 kW (COP of 1/100). LH2 will be cooled through a heat exchanger and circulated in the tank by a small pump.

Mass breakdown:

Supplementary solar panel surface:	20 kg (for 1000 W)
Cryocooler:	30 kg
Radiator (330 W / m^2):	30 kg (3 m^2)
Exchanger (in LH2 tank), plumbing:	20 kg
Total:	100 kg

The cryorefrigerator will use preferably a two stages scheme with the first stage providing a temperature of 90 K (it could be used also to cool the LOX tank) and the second stage 20 K.
The figure 4 gives a schematic diagram of the two stages cryo cooler.

4. LOX TANK INTEGRATION IN SMALL SPACECRAFT:

Most spacecraft present use a bipropellant mass around 1.5 ton (e. g. ROSETTA). The LOX / LH2 LCCP concept thermal control becomes more difficult since the surface / volume ratio is higher.
A LOX tank with a double walled vacuum insulation is an attractive option despite the increased tank mass. The fuel (light hydrocarbon, e.g. propane, propene), combined with LOX provides an interesting Isp (close to 360 s instead of 320 s for MMH / NTO).
The thermal losses in the vacuum insulated tank are reduced to 10 W.
The vacuum insulation offer another advantage: the LOX tank can be filled before erecting the spacecraft on top of the launcher. No launch pad servicing is needed.

This is especially true for two cases:

- Replacement of MMH / NTO by green propellants for medium robotic missions,
- On board supply of oxygen for a fuel cell powered Martian Rover.

Small quantities can be also stored in liquid form with double wall vacuum insulation like on SSHeL of ARIANE 5 (inducing practically no weight penalty).
Example:

➢	MMH / N2O4 mass	= 1500	kg	(very close to ROSETTA)
➢	go.Isp	= 3138	m/s	
➢	Total impulse	= 4.707	MN.s	
➢	LOX / C3H6 go.Isp	= 3580	m/s	
➢	LOX / C3H6 mass	= 1314.9	kg	
➢	(LOX mass)	= 928	kg	
➢	Delta vacuum insulation	= 185	kg	
Total		= 1500.5	kg	

- i. e. same mass! -
Of course, a small cryorefrigerator is also needed to balance the heat flux in the insulation.

5. MISSIONS

5.1. Direct injection to Mars

The benefit provided by LCCP is linked to the very low dry mass of LCCP module. The specific impulse is slightly lower than the one of VINCI (440 s versus 465 s) but this is more than more compensated by the reduced dry mass (900 kg instead of 5500 kg).
The figure 5 shows the injection procedure: the orbit apogee is raised by successive firings at perigee until the perigee speed is greater than 10.7 km/s. A last firing provides a ΔV of at least 600 m/s in order to insure injection into desired interplanetary orbit.

ARIANE 5 ECA:
Due to trajectory constraints, the direct injection to Mars provides a 3100 kg payload. Starting from a sub GTO orbit, LCCP can inject 7000 kg toward Mars.

ARIANE 5 ECB:
The payload mass toward Mars is increased from 7500 kg (direct injection with ECB and two VINCI engines firings) to 9200 kg (10 200 kg total mass including LCCP module).

5.2. Mars orbiter and lander

ARIANE 5 ECA
GTO (AR5 ECA) to Mars surface with a single stage Delta V = 2.80 km/s
landed mass: 4392 kg instead of 3406 kg (storable)

ARIANE 5 ECB:
After an insertion firing of 900 m/s, the mass in elliptical Mars orbit is 8300 kg. Taking into account the dry mass of LCCP stage, the solar panel and active refrigeration mass this leaves a useful mass of 7000 kg. Taking into account a mass percentage of 25 % for the entry equipment (heat shield, parachute, retrorocket) the landed mass is **5250 kg**.

This enables a Mars Sample Return mission with direct return to Earth. The return module can have a weight in excess of 4000 kg in the from of a two stages LOX – HC rocket enabling to provide a total Delta V of 5.3 km/s sufficient to return directly to Earth.

6. CONCLUSION:

LCCP provides a very high performance increase for planetary missions. Direct injection is a first opportunity.

Using the same stage with active refrigeration will enable an even larger payload gain.

LOX / HC option is very easy to develop with existing European technology, while a high efficiency, 1 kWe, 20 K active refrigerator will require more development. Consequently, LOX / HC could constitute a first step toward the use of full cryogenic propellants for long duration robotic missions. It could be especially useful for a direct Mars Sample Return scenario, the storage of LOX with active refrigeration being fairly easy.

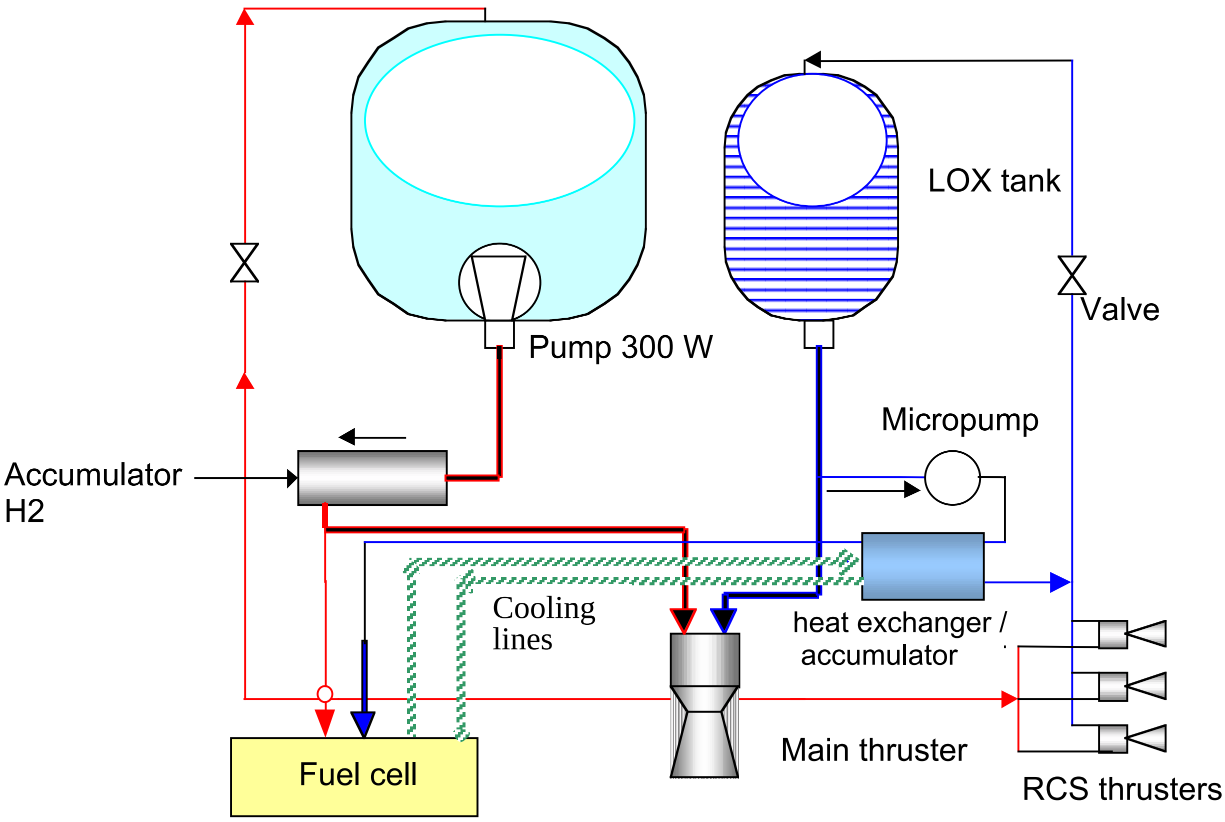

Figure 1: schematic diagram of LCCP

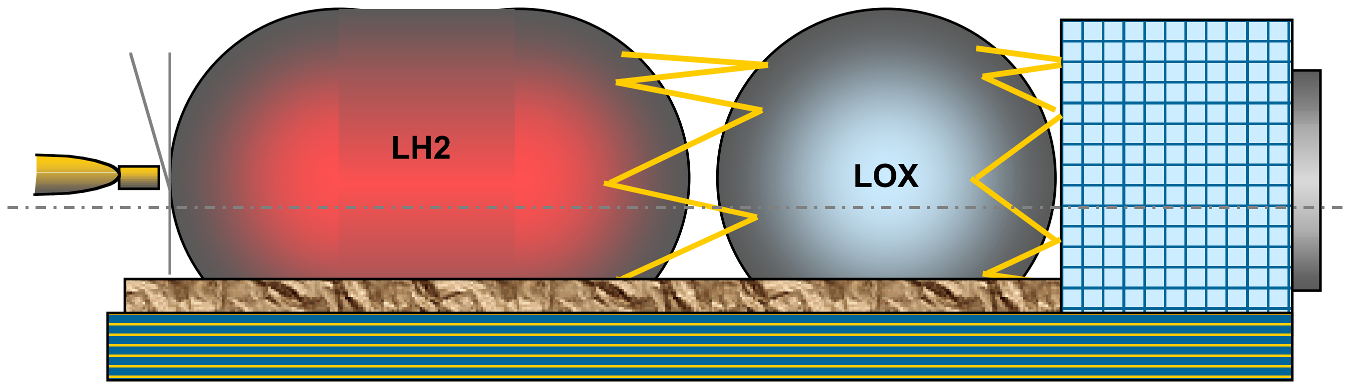

Figure 2: example of propellant tanks integration in a host spacecraft

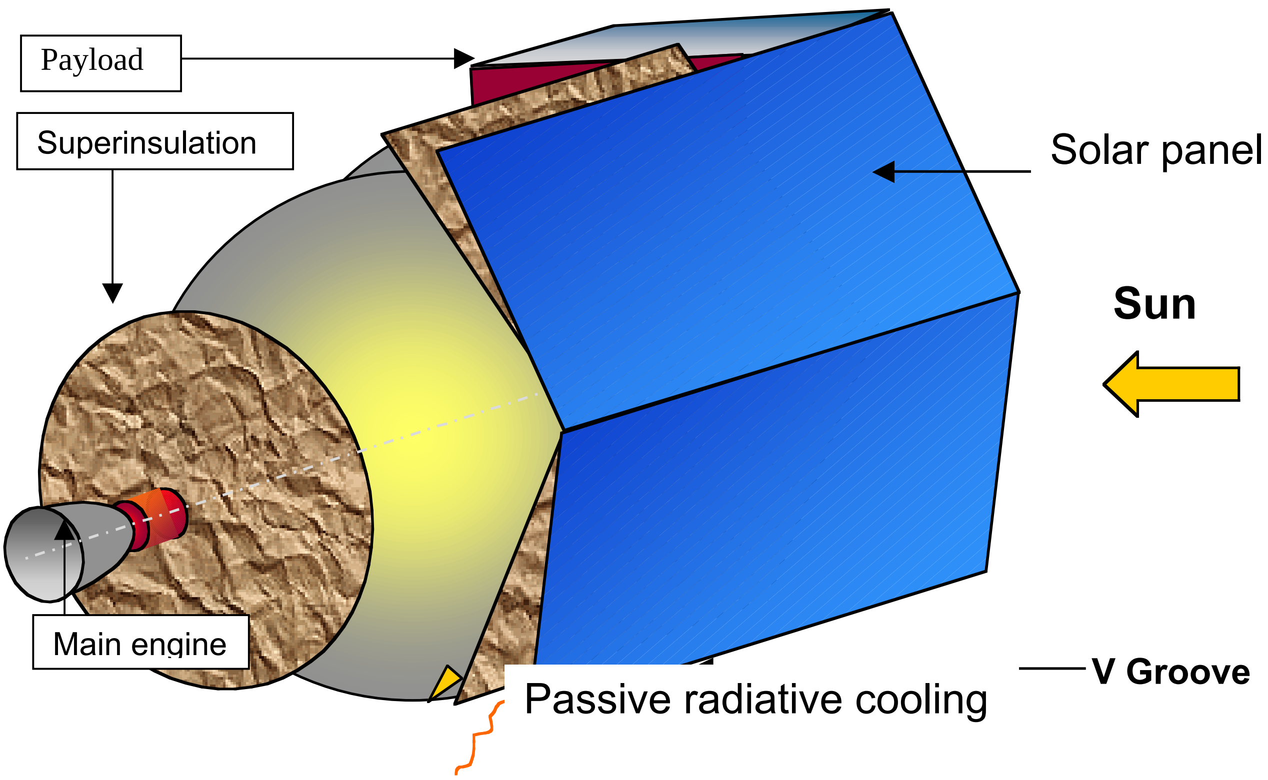

Figure 3: Radiative Thermal design features enabling to lower the thermal flux on tanks

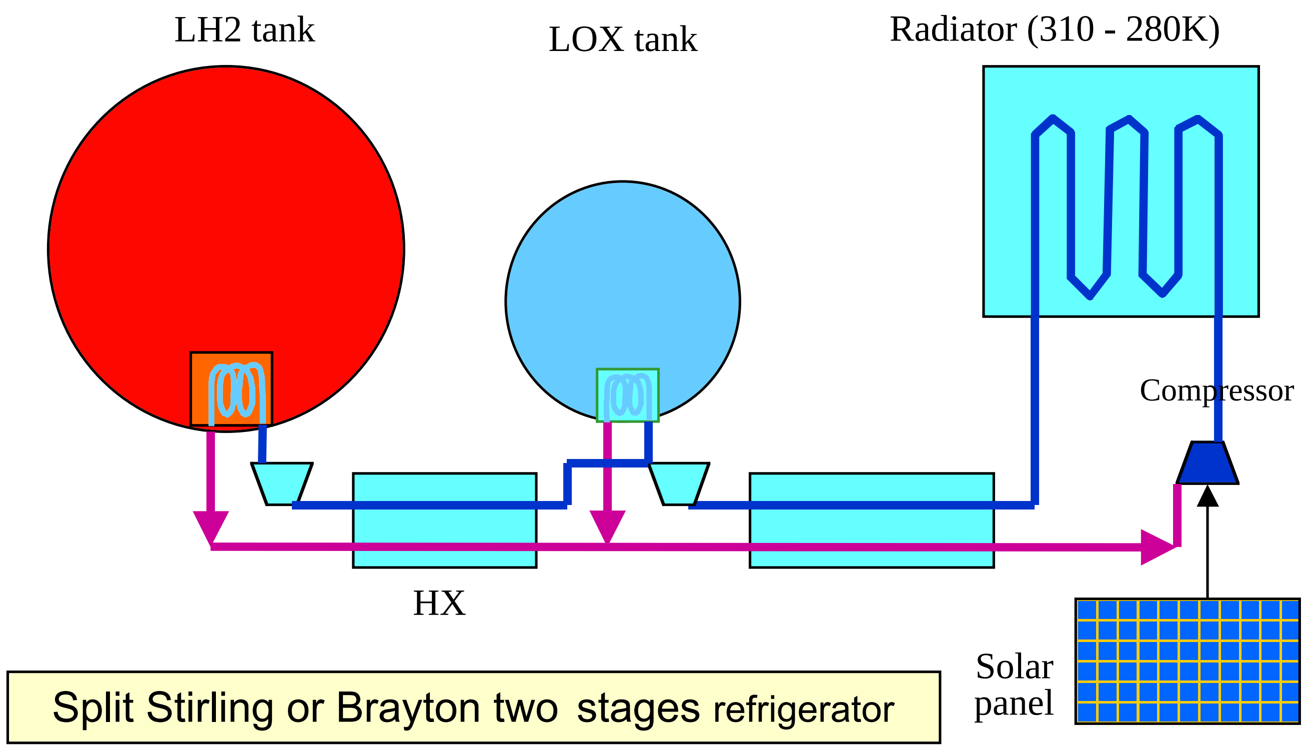

Figure 4: active refrigeration schematic diagram

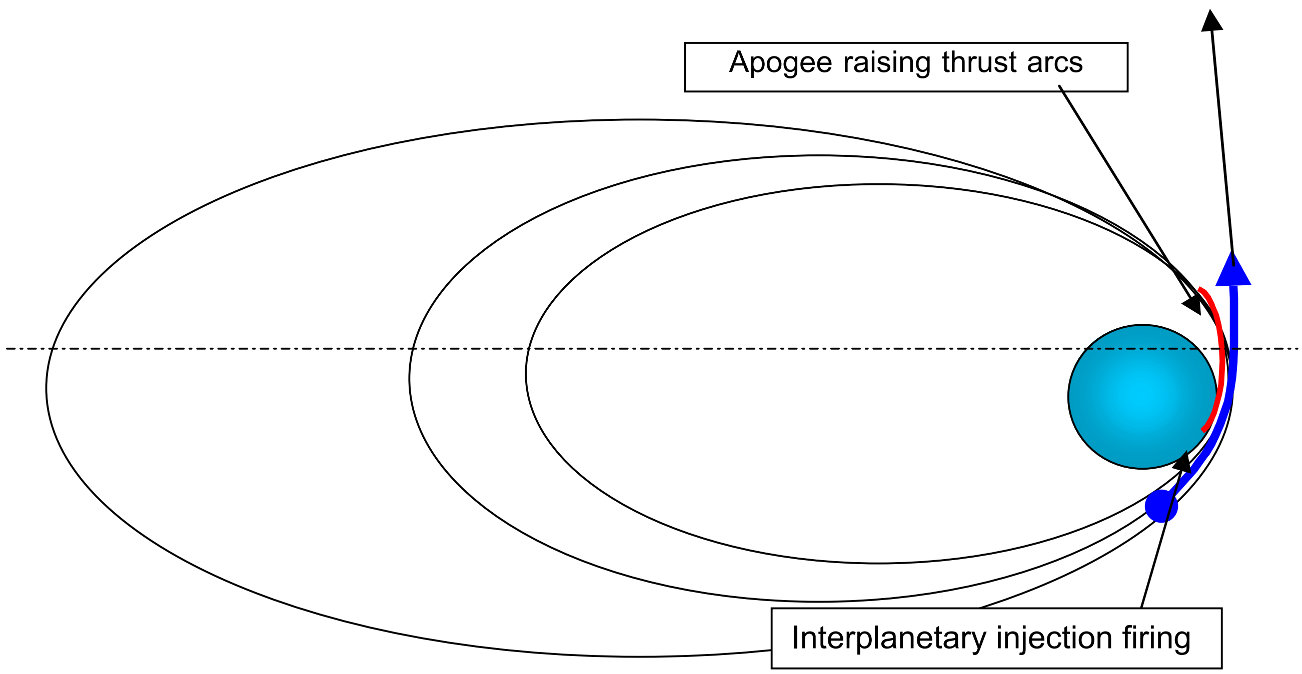

Figure 5: Multiple firing sequence for interplanetary injection: